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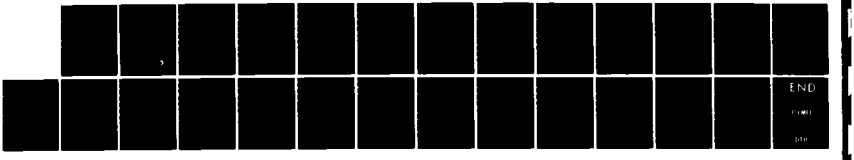

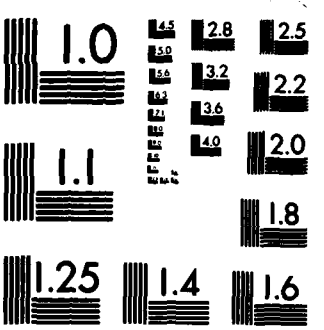

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STRUCTURES REPORT 403 / MATERIALS REPORT 116

ON THE EFFECTS OF DELAMINATION DAMAGE IN FIBRE COMPOSITE LAMINATES

by

R. JONES, J. PAUL and W. BROUGHTON

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SUMMARY

This paper describes the results of a numerical investigation into the effects of delamination and impact damage on the compressive strength of graphite epoxy laminates and bonded metal-to-composite joints. For the laminates considered it is shown that as the size of the damage increases a stage is reached after which any further significant increase in the damage results in only a relatively small decrease in the residual compressive strength.

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NOTATION

x, y, z	Cartesian coordinates
r, θ, z	Cylindrical polar coordinates
$\sigma_{ij}, \epsilon_{ij}$	Stress and strain components
G	Energy release rate
J	The J integral
K_1	Mode 1 stress intensity factor
σ_u	Unnotched tensile strength
d_o	Critical damage zone size
dw/dv	Strain energy density
r_c	Critical core-zone size
S	Strain energy density factor
ϵ_f	Far field failure strain

1. INTRODUCTION

Delamination damage in fibre composite materials may occur due to a variety of reasons, such as low energy impact or manufacturing defects. The presence of delamination damage is of major concern in the vicinity of bonded joints and in compressively loaded components where damage may grow under fatigue loading by out-of-plane distortion.

An early study into delamination growth arose from the B-1 composite development program [1]. This showed that delaminations can significantly reduce the fatigue life, and the residual compressive strength, for a compression dominated fatigue load spectrum. These effects have been confirmed in a series of recent articles [2, 3, 4, 5].

The present paper forms part of a joint investigation into delamination damage currently underway at both the Aeronautical Research Laboratories, Australia and the Royal Aircraft Establishment, England. In this work a three-dimensional finite element analysis is performed in order to understand the mechanisms involved in delamination growth, and subsequent failure, under compressive loading. This shows that as the size of damage increases a stage is reached after which any further increase in the damage produces only a relatively small decrease in the residual compressive strength.

2. LAMINATE CONFIGURATION

After examining the fracture surfaces of a quasi-isotropic laminate, which had failed under a compression fatigue spectrum, it was decided to test a series of laminates which contained a teflon disk between the second and third plies. The laminates were graphite epoxy with the following lay up:

(i) Type A: $(0/\pm 45/90)_{3S}$

(ii) Type B: $(0/\pm 45/90)_{4S}$

(iii) Type C: $(\pm 45/0/90/0)_{2S}$

The coupons constructed from laminate type A were 76·2 mm long, and 101·6 mm wide and contained a simulated delamination 25·4 mm × 19·05 mm between the 2nd and 3rd plies. The coupons constructed from laminate type B were 304·8 mm long and 101·6 mm wide with a delamination 38·1 mm × 25·4 mm in the same location. These two laminates were tested at the RAE and details of the experimental results can be found in [20].

The coupons constructed from laminate type C were 105 mm long and 45 mm wide and contained a variety of simulated delaminations and low-energy impact damage. These coupons were tested at ARL to determine the effect of the damage on the compressive strength of the laminate.

A detailed three-dimensional finite element analysis was performed on each set of coupon tests and the results of this investigation can be found in the following Sections.

3. FINITE ELEMENT ANALYSIS

The finite element analysis of complex three-dimensional delamination damage in laminated composites is particularly difficult. In general each ply must be modelled separately in order to obtain the correct values for the peel and the interlaminar shear stresses around the

delamination. In addition a fine mesh is required around the front of the delamination in order to correctly model the stress singularity. This results in a very large numerical model. In order to reduce the total number of degrees-of-freedom it is tempting to use eight-noded isoparametric bricks. Indeed this approach was used in [6, 7]. Unfortunately these elements cannot model the significant bending stresses which arise in the delamination problem [8]. Furthermore, since modelling is taking place at the ply level the elements have very large aspect ratios. This gives rise to problems of numerical ill-conditioning. Again, using the eight-noded bricks it is not possible to improve the conditioning of the problem. In general these elements should not be used with aspect ratios greater than five to one.

Consequently under no circumstances should eight-noded brick elements be used to model delamination problems in fibre composites.

The present investigation was done in double precision using twenty-noded isoparametric elements and a directionally reduced integration scheme with $2\times2\times3$ Gaussian quadrature points, with 3 points being used through the thickness of the ply. This integration scheme is described in more detail in [9]. However, as mentioned above, if each ply is modelled separately the numerical model becomes excessively large and so a new super-element was developed.

In this paper the plies above the delamination are modelled separately as are the two plies below the delamination. The remaining plies are treated as a super element with the displacements varying quadratically in the local isoparametric coordinate system, as in Reissner thick-plate theory. With this approach the stiffness matrix for each super-element can still be written in the conventional form, viz.:

$$K = \int\int\int BDB^{T}dv$$

$$= \sum_{i=1}^{N} \int\int\int_{V_i} BD_iB^{T}dv \qquad (1)$$

Here D_i is the elasticity matrix for the ith ply which has volume V_i.

There may be an arbitrary number of plies in a single super-element and thus unlike classical elements D varies throughout the thickness of the element. Details of the Gaussian quadrature required to integrate equation (1) correctly for an arbitrary number of plies is given in [9] and [10]. Such an approach significantly reduces the number of nodes and elements required and yet still allows for an accurate calculation of the stress field. However, even with this approach the problem consisted of six hundred of the twenty-noded isoparametric elements. As is now standard practice the mid-side nodes of the elements surrounding the delamination were moved to the quarter points in order to simulate the required singularity. It must be noted that when using eight-noded bricks as in [6, 7] it is not possible to represent the required stress singularity.

Having performed the stress analysis we must now decide on a criterion for assessing the severity of the delamination damage. Ideally we would like to evaluate the energy release rate G. This can be readily done for two-dimensional problems using either the method of virtual crack extension, as described in [11] for the edge delamination problem, or in the case of self-similar growth, by evaluating the J integral. For non self-similar growth $J \neq G$.

For problems which are not two dimensional there is no simple method for evaluating the energy release rate G without a prior knowledge of the way in which the delamination will grow. Furthermore whilst a line integral J_1^* has been developed [12], for three-dimensional fracture problems, this integral does not equal the true local energy release rate. This integral is defined by:

$$J_1^* = \lim_{\delta\to 0} \Phi \left(\tfrac{1}{2}\, \sigma_{ij} \frac{\partial ui}{\partial x} \cdot \mathbf{1} - T_i \frac{\partial u_i}{\partial x_j} \right) ds \qquad (2)$$

where the integration is along a contour of radius ρ normal to the front of the delamination. This integral coincides with the classical definition of J for 2-D problems, but for three-dimensional problems returns the local energy required for self-similar growth, i.e., circles into concentric circles, and not the true energy release rate. For metallic components under mode 1 fracture J_1^* is directly proportional to the local stress intensity factor K. Hence for metals J_1^* is a useful quantity. However, for our present problem the growth is mixed mode and non self-similar with the result that J_1^* is of questionable value.

Alternative approaches for assessing the severity of cracks and holes in fibre composite laminates have recently been developed [13, 14]. These methods are termed the point and average stress failure criteria. The point stress failure criterion assumes that failure will occur when the normal stress, σ_n, to the crack (or hole) at a distance d_o in front of the crack reaches the unnotched strength σ_u of the laminate, viz.:

$$\sigma_n|_{x = d_o} = \sigma_u \tag{3}$$

The quantity d_o is usually called the damage zone size and for graphite epoxy laminates is typically 0·9 mm (=0·038″).

The present paper uses the strain energy density approach [15, 16] to assess the delamination damage. This approach may be considered as an extension of the point stress failure criterion to allow for mixed mode failure.

Define the strain energy density in the usual fashion:

$$dw/dv = \tfrac{1}{2}\ \sigma_{ij}\ \epsilon_{ij} \tag{4}$$

Then for a two-dimensional problem the strain energy density approach has two basic hypotheses which apply for crack extension:

(1) The crack will grow in the direction $\theta = \theta_o$ of maximum potential energy density (viz.: minimum strain energy density).

(2) Failure occurs when the stress field at a distance r_c ahead of the crack (or hole) in the direction $\theta = \theta_o$ of the minimum strain energy density, is such that:

$$(dw/dv)_{\substack{r = r_c \\ \theta = \theta_o}} = (dw/dv)_c \tag{5}$$

Here $(dw/dv)_c$ is the value, at failure of the strain energy density of the undamaged laminate. In this formulation r_c plays a similar role to d_o, the damage zone size used in the point stress failure formulation. Indeed as a first approximation we can take $r_c \simeq d_o$. In this approach the strain energy density function S defined by

$$dw/dv = \frac{S}{r} + \text{higher order terms} \tag{6}$$

plays a central role. For mode 1 failure of orthotropic material S is proportional to the stress intensity factor K_1. Indeed the above hypothesis can be readily expressed in terms of S; viz.:

Failure occurs in the direction θ_o for which

$$\partial S/\partial\theta = 0 \text{ and } \partial^2 S/\partial\theta^2 > 0 \tag{7}$$

and when the load is such that

$$S|_{\theta = \theta_o} = S_c\ (=r_c\ (dw/dv)_c) \tag{8}$$

For three-dimensional damage we must first locate at each point along the damage front the local *minimum* of the strain energy density function $S_{min}(= r\, dw/dv_{min})$. Failure then initiates at the point along the front which has the maximum value of S_{min}.

For incremental growth each point along the damage front advances a distance r which is determined from the relationship

$$\frac{S_1}{r_1} = \frac{S_2}{r_2} = \ldots = \frac{Si}{r_1} = \left(\frac{dw}{dv}\right)_c$$

A more detailed description of this hypothesis including its applicability to mixed mode crack growth can be found in [15, 16].

4. NUMERICAL RESULTS

For the tests on laminates types A and B the finite element model yielded the maximum values of S_{min} along the lines AA' and BB', see Figure 1. Indeed the values* along these lines were relatively constant with the maximum value of S_{min} occurring at point D, approximately 3 mm below the centre line for specimens A and B. Specimen C, constructed from laminate type C, had the maximum value of S_{min} occurring exactly at the midsides of AA' and BB'.

The values of $S^* = (S_{min})_{max}$ for laminates type A and B are given in Table 1 for a uniform compressive strain of 0·004 applied to the ends of the specimen.

TABLE 1

Values of S*

Laminate type	S^*(MPa.mm)	ϵ_f
A	0·0033	0·0061
B	0·0022	0·0075

In order to estimate the failure load we need the critical value of S, i.e. S_c. One early value of S_c for an epoxy was given as 0·055 lb/in (= 0·0096 MPa.mm), but more recent work [16] has found that a value of $S_c = 0·044$ lb/in (= 0·0077 MPa.mm) is more representative. This value is for a Modulite II 5206 graphite expoy.

Using this value for S_c and assuming that the coupon behaviour remains linear elastic we can now estimate the strain required to cause failure which we will denote as ϵ_f. This value is given in Table 1.

It is particularly interesting to compare these predicted compressive failure strains with those given in [2] for a similar quasi-isotropic laminate with 32 plies, cf. laminate type B, and one with 24 plies; see Tables 2 and 3 respectively.

TABLE 2

Compressive failure strain ϵ_f for a 32-ply quasi-isotropic laminate [from (2)]

	Average Damage Area (mm)				
ϵ_f	426	523	587	671	929
	0·0077	0·0075	0·0073	0·0067	0·0075

* The values of S were obtained via an interpolation procedure which is described in the Appendix.

TABLE 3

Compressive failure strain ϵ_f for a 24-ply $(0/45/0_2/-45/0_2/45/0_2/-45/0)_S$ laminate [from (2)]

	Average Damage Area (mm)				
ϵ_f	458	574	645	703	955
	0·0060	0·0064	0·0057	0·0057	0·0060

As can be seen from Table 2 the failure strains predicted for laminate type B compare favourably with those measured in [2].

For the coupons constructed from laminates type C three different sizes of delaminations were modelled. The delaminated areas chosen were

(*a*) 1″ × 1″ (25·4 mm × 25·4 mm)

(*b*) 1″ × 1½″ (25·4 mm × 38·1 mm)

(*c*) 1″ × 2″ (25·4 mm × 50·8 mm)

In each case the value of $((dw/dv)_{min})_{max}$ occurred exactly at the centre of lines AA' and BB'. The corresponding values of $S^* = (S_{min})_{max}$ for a compressive strain of 0·004 are shown in Table 4 along with the predicted values for ϵ_f assuming as before that $S_c = 0{\cdot}044$ lb/in (= 0·0077 MPa.mm).

TABLE 4

	Delamination sizes (mm²)		
	645	967	1290
S^* (MPa.mm)	0·0036	0·0034	0·0033
ϵ_f	0·0057	0·0060	0·0061

Allowing for numerical error the value of S^*, and hence the failure strain ϵ_f, remains fairly constant as the delamination size increases. This phenomenon can be seen to occur in the experimental results given in [2] and summarized in Tables 2 and 3. Indeed this can also be seen in the experimental results given in [4].

In the coupon tests on laminate C a number of 6·35 mm, 15·8 mm and 25·4 mm diameter inclusions were tested. These specimens failed at far-field strains of approximately 5200 μ, 5100 μ and 4980 μ respectively. Whilst there is a slight reduction in the far-field failure strain in going from the 6·35 mm diameter inclusion to the 25·4 mm diameter inclusion, this increase in area resulted in only a 5% decrease in strength.

In addition to this set of tests a second series of tests was performed on impact-damaged specimens. The failure strain ϵ_f for these specimens is shown in Table 5.

TABLE 5

Compressive strength of impact damaged specimens

	Damaged area (mm^2)			
ϵ_f	0·0	38	195	314
	0·0079	0·0067	0·0058	0·0049

It thus appears that as the size of the damaged area increases the value of S^* $(= (S_{min})_{max})$ asymptotes to a constant value. As a result since failure occurs when $S^* = S_c$ the residual strength also asymptotes to a constant level as the damage area increases.

It is important to note that this phenomenon has also been observed in the compressive strength of composite laminates containing edge delaminations [11, 17, 18, 19]. Although the present paper has used only small deformation theory in attempt to include large deformation effects is currently underway. Indeed recent analytical work* by Professor D. C. Stouffer in the Department of Aerospace Engineering at the University of Cincinnatti has shown that, even when large deformations occur, the energy release rate, and hence ϵ_f, asymptotes to a constant value as the size of the delamination increases.

5. DAMAGED STEP-LAP JOINTS

Let us now examine the effect of delamination damage at a bonded step-lap joint. In order to understand the mechanisms involved a relatively simple joint configuration was considered; see Figure 2. The composite is a $(\pm 45/02)_{2S}$ graphite epoxy with the delamination occurring at the interface between the titanium step and the 0 degree.

The moduli of the graphite epoxy were taken to be

$$G_{13} = G_{12} = G_{23} = 5{\cdot}0 \text{ GPa},$$

$$E_{22} = E_{33} = 9{\cdot}5 \text{ GPa},$$

$$E_{11} = 141 \text{ GPa},$$

$$\nu_{12} = \nu_{13} = 0{\cdot}31,$$

$$\nu_{23} = 0{\cdot}021.$$

The titanium was assumed to have $E = 110$ GPa and $\nu = 0{\cdot}3$.

As before each ply was modelled separately using twenty-noded isoparametric bricks and the region around the crack front was modelled using the fifteen-noded isoparametric wedge elements with the mid points moved to the quarter points. A cross-section of the finite element mesh can be seen in Figure 3.

* Private communication.

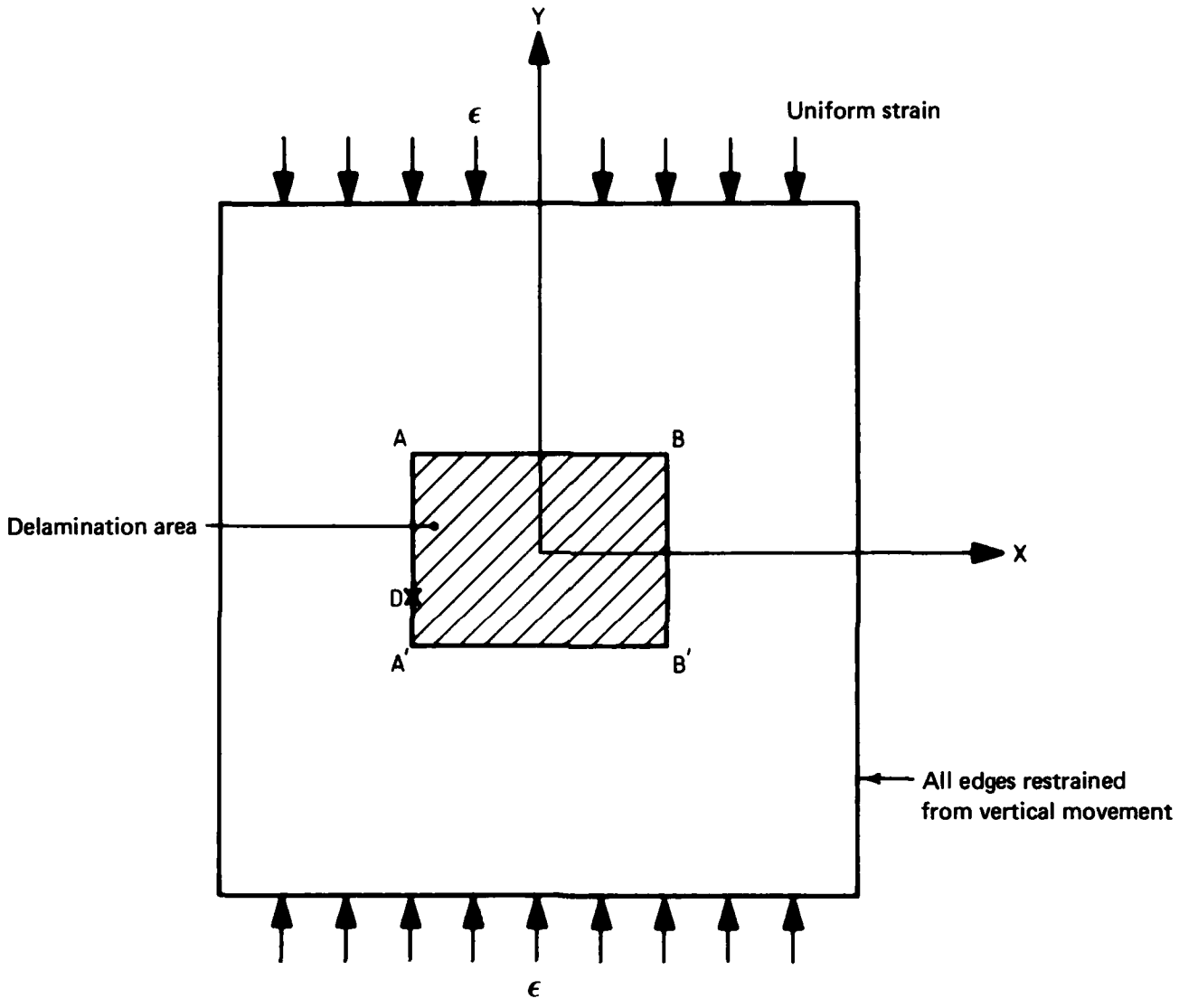
Y
ϵ
Uniform strain
A
B
Delamination area
X
D
A'
B'
All edges restrained
from vertical movement
ϵ

FIG. 1

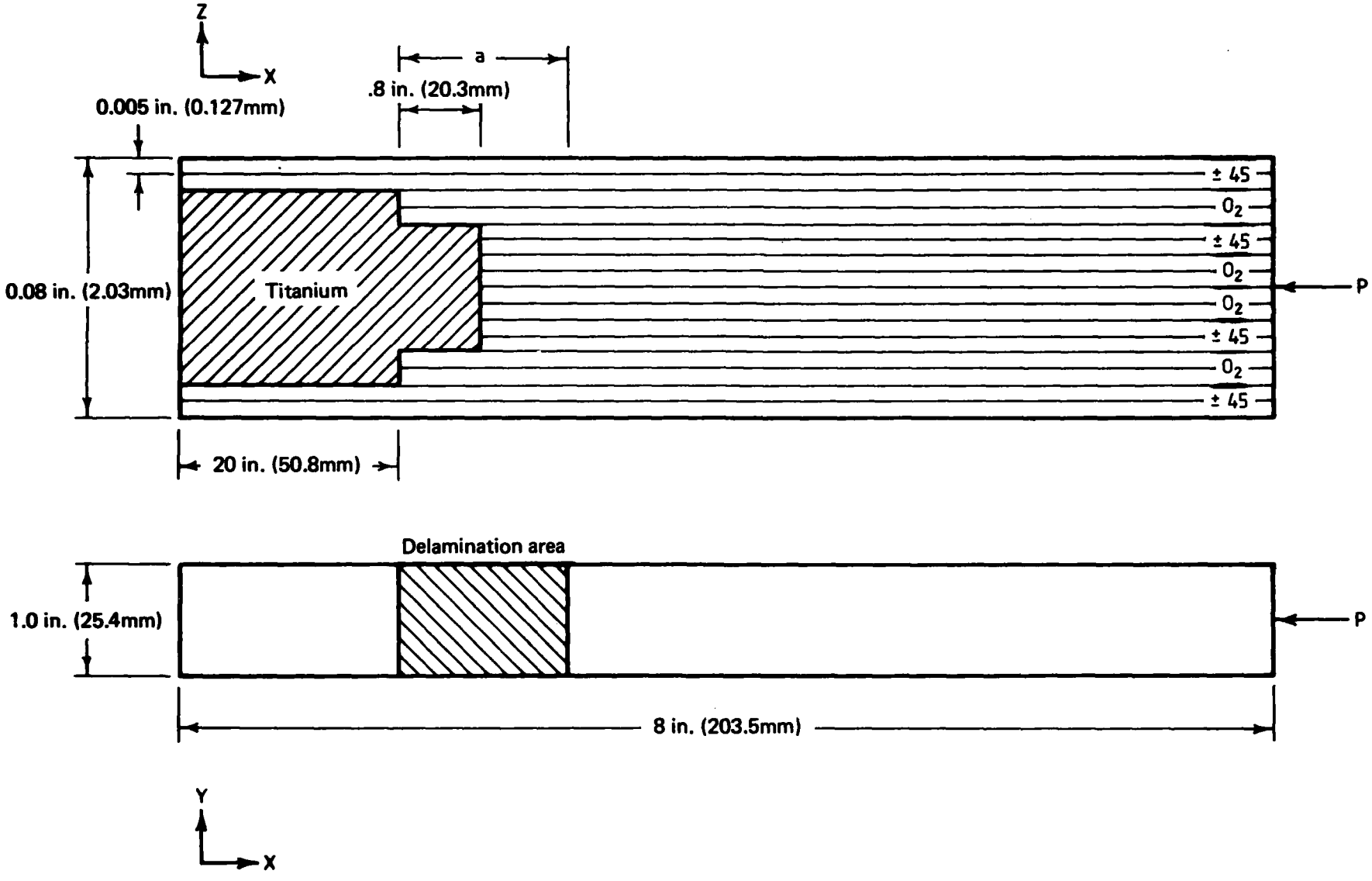

FIG. 2 PLY LAYUP AND DIMENSIONS

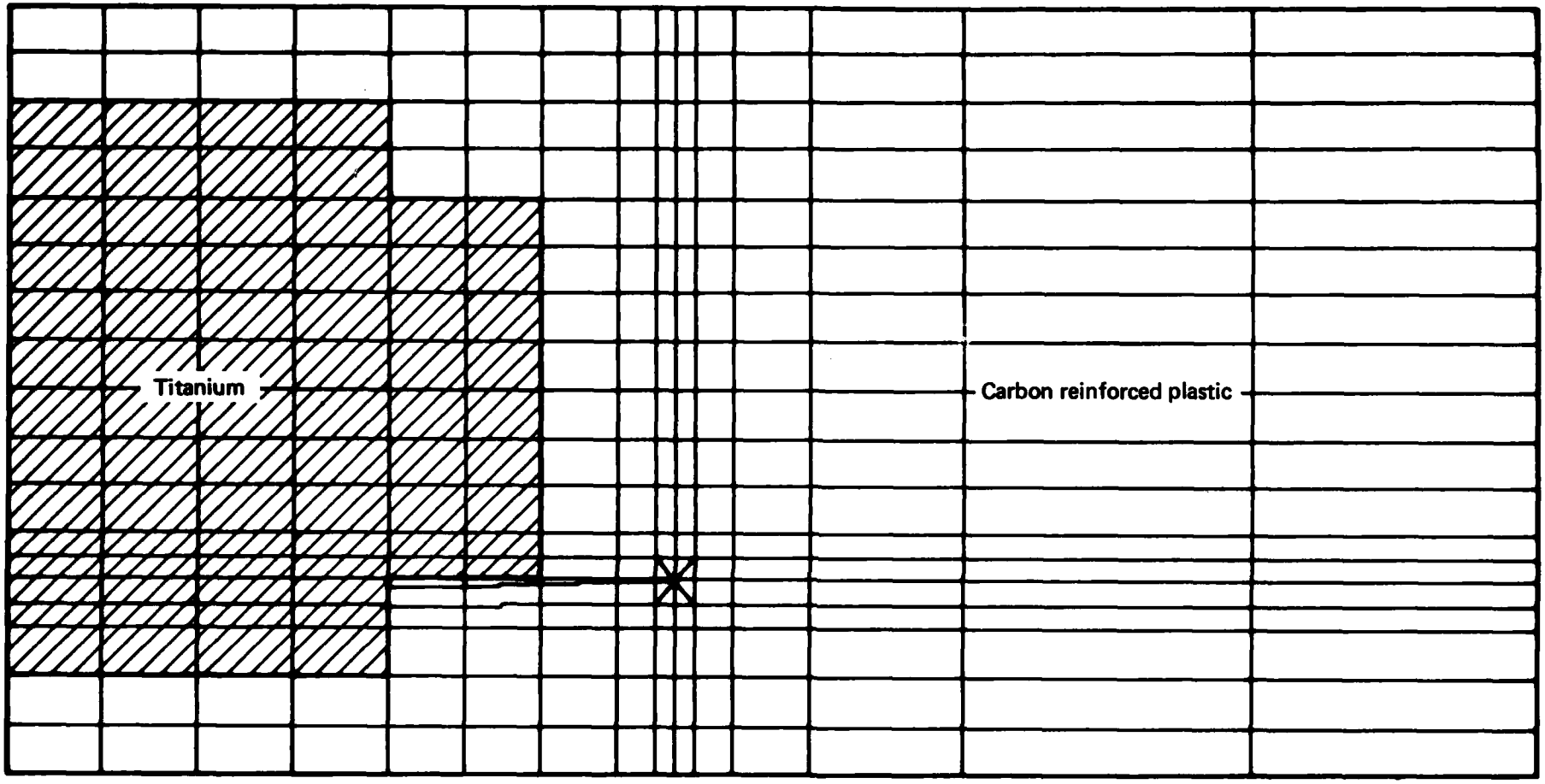

FIG 3 FINITE ELEMENT MESH X – SECTION

The delamination was found to close under applied compressive loading. As a result constraints were applied to the faces of the delamination so as to prevent the faces from crossing (i.e. overlapping). In this case the stresses and deformations were essentially two-dimensional so that the energy release rate G could be calculated from the relationship.

$$G = \frac{P^2}{2B} \frac{\partial \delta / P}{\partial a}$$

where P is the applied load B is the width of the specimen and δ is the movement of the load point.

The values of G thus calculated are given in Table 6 for various size delaminations. These values correspond to an applied compressive strain of 4000 μ.

TABLE 6

	Delaminated area (mm²)					
	31·75	38·1	41·27	44·45	47·6	50·8
G(MPa.mm)	0·0254	0·0284	0·029	0·0302	0·0314	0·0326

Here we see that going from a delamination of 31·75 mm² to 50·8 mm², a 60% increase in the size of the damage, results in an increase in G of only 28%. This corresponds to a decrease in the residual strength of only 13%.

6. CONCLUSION

In this work we have found that for delamination damage, to both composite laminates and metal-to-composite joints, as the size of the damage increases a stage is reached after which any further significant increase in the damage produces only a small decrease in the residual compressive strength.

The next phase of this investigation involves detailed specimen testing in order to confirm the numerical results.

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APPENDIX

Consider two points at distances r_1 and r_2, such that $r_1 > r_2$, and which lie in a straight line in the direction of crack (i.e., delamination) growth. Both r_1 and r_2 are chosen to be much smaller than the crack length. In the present work we have also chosen $r_1 = d_o = 0{\cdot}9$ mm ($= 0{\cdot}038''$). Then from equation (6) we see that

$$\left(\frac{dw}{dv}\right)_{r_1} - \left(\frac{dw}{dv}\right)_{r_2} = S\left(\frac{1}{r_1} - \frac{1}{r_2}\right) \tag{A1}$$

which gives the value of S as

$$S = [(dw/dv)_{r_1} - (dw/dv)_{r_2}]/(1/r_1 - 1/r_2) \tag{A2}$$

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DOCUMENT CONTROL DATA

1. a. AR No.	1. b. Establishment No.	2. Document Date	3. Task No.
AR-003-927	ARL-STRUC-R-403/ ARL-MAT-R-116	June 1984	AIR 80/126

4. Title	5. Security	6. No. Pages
ON THE EFFECTS OF DELAMINATION DAMAGE IN FIBRE COMPOSITE LAMINATES	a. document: Unclassified b. title: U c. abstract: U	14 7. No. Refs: 20

8. Author(s)	9. Downgrading Instructions
R. Jones, J. Paul and W. Broughton	

10. Corporate Author and Address	11. Authority (*as appropriate*)
Aeronautical Research Laboratories P.O. Box 4331, Melbourne, Victoria, 3001	a. Sponsor c. Downgrading b. Security d. Approval (a) AIR 80/126 (b) Dept. of Defence (Air Force Office)

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14. Descriptors	15. COSATI Group
Damage	11040
Delaminating	11130
Fibre composites	
Fibre laminates	

16. Abstract

This paper describes the results of a numerical investigation into the effects of delamination and impact damage on the compressive strength of graphite epoxy laminates and bonded metal-to-composite joints. For the laminates considered it is shown that, as the size of the damage increases, a stage is reached after which any further significant increase in the damage results in only a relatively small decrease in the residual compressive strength.

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16. Abstract (*Contd*)

17. Imprint

Aeronautical Research Laboratories, Melbourne.

18. Document Series and Number	19. Cost Code	20. Type of Report and Period Covered
Structures Report 403/ Materials Report 116	214705	—

21. Computer Programs Used

—

22. Establishment File Ref(s)

—

END

FILMED

4-85

DTIC